Development of a Flight Test Maneuver Autopilot for an F-15 Aircraft

Gurbux S. Alag and Eugene L. Duke

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Gurbux S. Alag
Western Michigan University, Kalamazoo, Michigan
Eugene L. Duke
Ames Research Center, Dryden Flight Research Facility, Edwards, California

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Gurbux S. Alag
Western Michigan University
Kalamazoo, Michigan

and

Eugene L. Duke
NASA Ames Research Center
Dryden Flight Research Facility
Edwards, California

An autopilot can be used to provide precise
control to meet the demanding requirements of flight
research maneuvers with high-performance aircraft.
This paper presents the development of control laws
for a flight test maneuver autopilot for an F-15
aircraft. A linear quadratic regulator approach is
used to develop the control laws within the context
of flight test maneuver requirements by treating the
maneuver as a finite time tracking problem with regu-
lation of state rates. Results are presented to show
the effectiveness of the controller in insuring accept-
able aircraft performance during a maneuver.

Introduction

Conventional piloting techniques are often inade-
quate to meet the demanding requirements of flight
research maneuvers with high-performance aircraft.
These maneuvers frequently require precise control of
onset rates in extreme flight conditions. Thus, the
pilot may be trying to control an aircraft at high
angles of attack and high g's while attempting to
increase normal acceleration at a prescribed rate
through a maneuver specified to the very limits of the
accuracy of the cockpit instruments.

A new flight test technique to aid the pilot
during these maneuvers was developed at the Dryden
Flight Research Facility of the NASA Ames Research
Center (Ames-Dryden).[1] The essence of this tech-
nique is the application of an autopilot to provide
precise control during the required flight test
maneuvers. The flight test maneuver autopilot (FTMAP)
is designed to provide precise, repeatable control of
a high-performance aircraft during certain prescribed
maneuvers so that a large quantity of data can be
obtained in a minimum of flight time.

The FTMAP can be used for various maneuvers, such
as straight-and-level flight, level accelerations and
decelerations, pushover pullups, excess-thrust windup
turns, and thrust-limited turns. Each of these maneu-
vers comprise tracking certain states of the aircraft,
holding certain states within prescribed values, and
maintaining constraints on the derivatives of the
states. For example, an excess-thrust windup turn is
performed at constant altitude and Mach number with
the angle of attack increasing at a specified rate to
the final angle of attack.

This paper presents the development of the FTMAP
control laws within the context of flight test maneu-
ver requirements. The FTMAP design represents a well-
defined and completely specified problem against which

various design methodologies can be tested. Not only
does this problem represent a real-world situation,
but the FTMAP control law design problem also forces
consideration of a significant design issue in air-
craft controls — controlling a plant that changes as
vehicle attitude or flight conditions change.

Problem Formulation

The functional capability of the FTMAP is derived
from its ability to generate control laws required
to execute a maneuver. The FTMAP control laws are
developed using the Ames-Dryden detailed nonlinear
aerodynamic model of the F-15 aircraft. The model
is linearized by trimming the aircraft at the desired
flight condition and deriving linear models by numeri-
cal perturbation. The linear model may be represented
in the standard state equation form as

$$\dot{x} = Ax + Bu \qquad (1)$$

$$y = Hx \qquad (2)$$

where the states x and inputs u represent perturbation
around nominal trim values $\bar{x}$ and $\bar{u}$, respectively.
Also, y is the output and A, B, and H are matrices of
appropriate dimensions.

In order to perform a certain maneuver, a control
algorithm is developed to take the aircraft from a
certain specified initial trim state to another spec-
ified state. The FTMAP control law design can be
formulated as an optimal tracking problem, in which
it is desired to track a constant final state in
finite time. It is also necessary to regulate the
state rate in certain maneuvers.

Multivariable linear quadratic control theory is
a powerful tool for the development of the flight test
controller. The control design problem selects inputs
u to drive the perturbation states x to the desired
final state by optimizing a quadratic performance
index. Because control of the derivatives of states
is also desired, additional terms are added to the
performance index and the resultant control laws are
computed directly from the performance index.

Control Law Synthesis

An optimal tracking problem may be formulated as
follows. Reiterating Eqs. 1 and 2, the system is
given as

$$\dot{x} = Ax + Bu$$

$$y = Hx$$

It is desired to minimize a performance index of the form

$$J = 1/2 \int_0^T [e(t)^T Q e(t) + u(t)^T R u(t)] \, dt \quad (3)$$

where $e(t)$ is equal to $z(t) - y(t)$, $z(t)$ is the desired state to be tracked and is a known function, and Q and R are weighting matrices.

To track a constant final state z and also to be able to regulate the derivatives of the states, an additional term is added to the performance index to be minimized. The performance index may be written as

$$J = 1/2 \int_0^T [e(t)^T Q e(t) + u(t)^T R u(t)$$

$$+ (E\dot{x} - E_0)^T S(E\dot{x} - E_0)] \, dt \quad (4)$$

Here,

$$e = z - y(t)$$
$$Q \geqslant 0$$
$$R > 0$$
$$S \geqslant 0$$

Also, z is the desired constant final state, and E_0 is the desired state rate.

If the performance index is minimized by using Lagrange multiplier, the optimal control u^* that will derive the aircraft to the desired terminal condition is

$$u^* = -(R + B^T E^T SEB)^{-1}[B^T P(t)$$

$$+ B^T E^T SEAx - B^T E^T SE_0]$$

$$= -R_1^{-1}[B^T P(t) + B^T E^T SEAx - B^T E^T SE_0] \quad (5)$$

where

$$R_1 = (R + B^T E^T SEB)$$

$$P(t) = K(t)x(t) - g(t)$$

and K and g are obtained from the solution of the following equations using appropriate terminal conditions.[2] Hence,

$$\dot{K} + \tilde{A}^T K + K\tilde{A} - K\tilde{E}K + \tilde{Q} = 0 \quad (6)$$

$$\dot{g}(t) + (\tilde{A} - \tilde{E}K)^T g(t) + H^T \tilde{Q} Z - LE_0 = 0 \quad (7)$$

where

$$\tilde{A} = A - BR_1^{-1}B^T E^T SEA$$

$$\tilde{E} = BR_1^{-1}B^T$$

$$\tilde{Q} = H^T QH + A^T E^T SEA - A^T E^T SEBR_1^{-1}B^T E^T SEA$$

$$L = -A^T E^T S + A^T E^T SEBR_1^{-1}B^T E^T S + K^T B^T R_1^{-1}B^T E^T S$$

However, the optimal control u^* need not be derived by a finite time formulation but may be ob-

tained by employing steady-state control laws. This is equivalent to assuming that the $z(T)$ persists as a constant value for a long period. With the foregoing assumption, K is a constant matrix that satisfies the algebraic Riccati equation and $\dot{g}(t) = 0$. Thus, the simplified control law becomes

$$u^*(t) = -C_1 x + C_2 z + C_3 E_0 \quad (8)$$

where

$$C_1 = R_1^{-1}B^T K + R_1^{-1}B^T E^T SEA$$

$$C_2 = -R_1^{-1}B^T L_1^{-1}H^T Q$$

$$C_3 = R_1^{-1}(B^T L_1^{-1}L + B^T E^T SE)$$

$$L_1 = (\tilde{A} - \tilde{E}K)^T$$

Application of Control Laws

The linearized equations of motion of an F-15 aircraft are given by Eq. 1

$$\dot{x} = Ax + Bu$$

where

$$x = \begin{bmatrix} v \\ \alpha \\ q \\ \theta \\ h \\ \beta \\ p \\ r \\ \phi \end{bmatrix} \begin{matrix} = \text{velocity, ft/sec} \\ = \text{angle of attack, deg} \\ = \text{pitch rate, deg/sec} \\ = \text{pitch angle, deg} \\ = \text{altitude, ft} \\ = \text{sideslip angle, deg} \\ = \text{roll rate, deg/sec} \\ = \text{yaw rate, deg/sec} \\ = \text{roll angle, deg} \end{matrix}$$

$$u = \begin{bmatrix} \delta_a \\ \delta_e \\ \delta_r \\ \delta_T \end{bmatrix} \begin{matrix} = \text{aileron deflection} \\ = \text{elevator deflection} \\ = \text{rudder deflection} \\ = \text{throttle displacement} \end{matrix}$$

For a flight condition corresponding to an altitude of 20,000 ft and Mach 0.8, the matrices A and B are as follows. For matrix A, columns 1 to 9 are

$$\begin{bmatrix}
-0.0108 & 24.1966 & 0.0000 & -32.1129 \\
-0.0001 & -1.0942 & 1.0000 & 0.0001 \\
-0.0001 & -3.2862 & -2.1922 & 0.0009 \\
0.0000 & 0.0000 & 1.0000 & 0.0000 \\
0.0000 & -829.5390 & 0.0000 & 829.5390 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000
\end{bmatrix}$$

$$\begin{bmatrix}
0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 \\
0.0000 & -0.2337 & 0.0358 & -0.9994 & 0.0387 \\
0.0000 & -40.0103 & -2.1420 & 1.2406 & 0.0000 \\
0.0000 & 9.0098 & -0.0340 & -0.6040 & 0.0000 \\
0.0000 & 0.0000 & 1.0000 & 0.0358 & 0.0000
\end{bmatrix}$$

Matrix B is given by

$$\begin{bmatrix} 0.0000 & -1.0734 & 0.0000 & 0.3792 \\ 0.0000 & -0.1504 & 0.0000 & 0.0000 \\ 0.0000 & -16.1223 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 \\ -0.0022 & 0.0000 & -0.0388 & 0.0000 \\ 13.5934 & 0.0000 & -1.4674 & 0.0000 \\ 0.1488 & 0.0000 & -4.5577 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 \end{bmatrix}$$

Three sets of trajectories show the results of the application of developed control laws to the aircraft. In the first case, it is necessary to hold the aircraft at trim conditions of the specified flight condition. The flight condition corresponds to an altitude of 20,000 ft and a Mach number of 0.8. The aircraft is in straight-and-level flight. The control law design is intended to maintain the aircraft at the specified altitude and velocity.

For this first case, figures 1 and 2 show the variation in velocity and altitude, respectively, when the aircraft is required to be flown straight and level at a given altitude and Mach number. Figures 3 and 4 show the corresponding variations in the elevator deflection and throttle displacement, respectively. The aircraft is held to the trim values by the controller designed.

In the second case, the flight condition corresponds to an altitude of 20,000 ft and the initial Mach number is 0.5. The aircraft is to be accelerated to Mach 0.8 while it is held at the specified altitude. Because of the variation of the parameters while the plane accelerates, the feedback gains were updated midway through the maneuver. Figures 5 and 6 show the variations of velocity and altitude as the aircraft undergoes the maneuver. The response was not considered satisfactory because of spikes observed at the time of update of gains. Figures 7 and 8 show the variations of the velocity and altitude for the same maneuver when the aircraft is initialized to the flight condition corresponding to Mach 0.8.

In the third case, the maneuver that was performed was the same as in the second case, except that a constraint was placed on the state rate. It is desired to maintain the Mach rate at 0.005 Mach per sec. Figures 9 and 10 show the variations in the velocity and altitude as the aircraft goes through the maneuver. The response obtained in this third case was considered satisfactory.

Concluding Remarks

This paper presents a synthesis technique that is applicable for control of an aircraft undergoing a specified maneuver. It assumes that a given maneuver can be modeled as a state trajectory to be tracked. The technique that is described uses the optimal regulator approach to the trajectory control problem. Results are presented for trajectory tracking for the aircraft undergoing altitude-hold level acceleration.

For more complex maneuvers, it is necessary to generate a set of state and control histories to serve as commands for the flight test maneuver autopilot (FTMAP), using a data base that consists of trim conditions. The maneuver model can be divided into a smaller set of state trajectories for suitable updating of aircraft parameters and gain scheduling. The development has the potential of simultaneously controlling multiple parameters to demanding tolerances.

References

[1]Duke, E.L., Jones, F.P., and Roncoli, R.B., "Development of a Flight Test Maneuver Autopilot for a Highly Maneuverable Aircraft," AIAA 21st Aerospace Sciences Meeting, Reno, Nev., Jan. 1983.

[2]Alag, G.S., and Duke, E.L., "Development of Control Laws for a Flight Test Maneuver Autopilot for an F-15 Aircraft," American Control Conference Proc., Boston, Mass., June 1985, pp. 1209-1210.
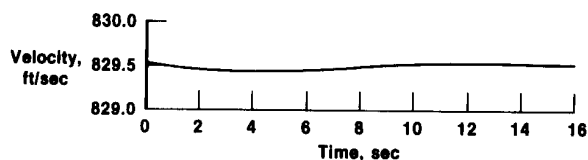
Fig. 1 *Variations in velocity with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude*
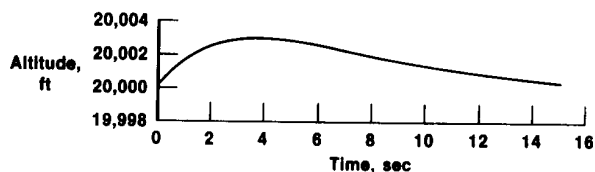

Fig. 2 *Variations in altitude with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude*
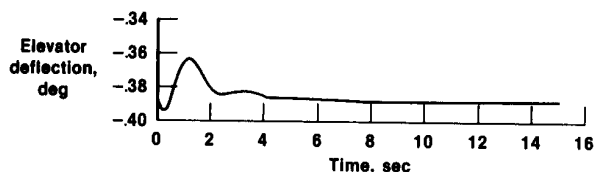
Fig. 3 *Variations in elevator deflection with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude*
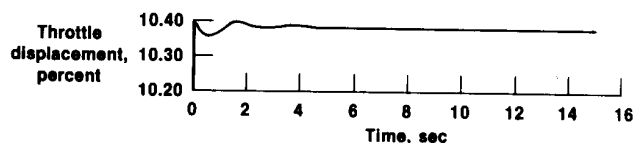
Fig. 4 *Variations in throttle displacement with time for straight-and-level flight condition of Mach 0.8 and 20,000-ft altitude*
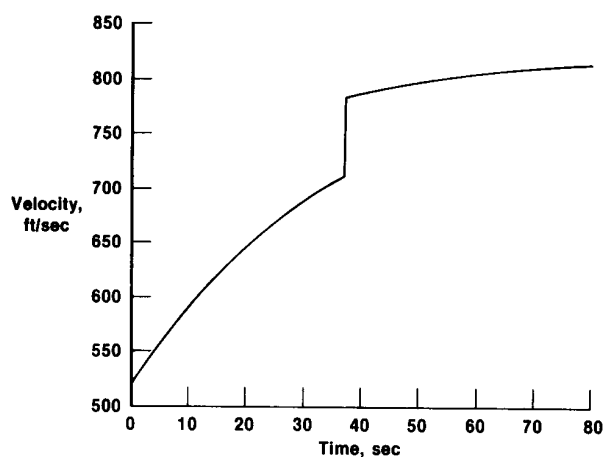
Fig. 5 *Variations in velocity with time for flight condition of 20,000-ft altitude with initial Mach 0.5 accelerated to Mach 0.8 (gains updated midway through maneuver)*
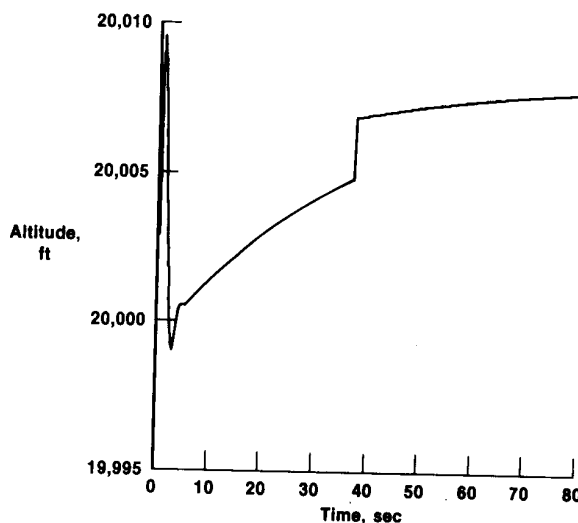
Fig. 6 *Variations in altitude with time for flight condition of 20,000-ft with initial Mach 0.5 accelerated to Mach 0.8 (gains updated midway through maneuver)*

4
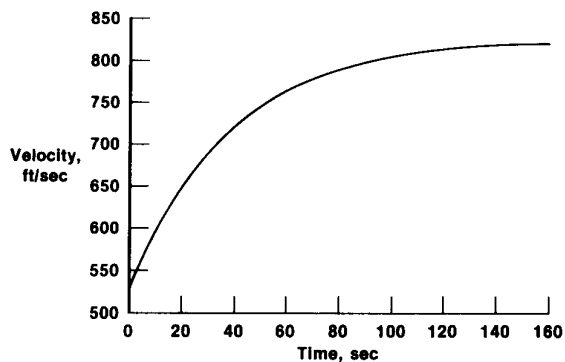
Fig. 7 Variations in velocity with time with aircraft parameters initialized to Mach 0.8 at 20,000-ft altitude
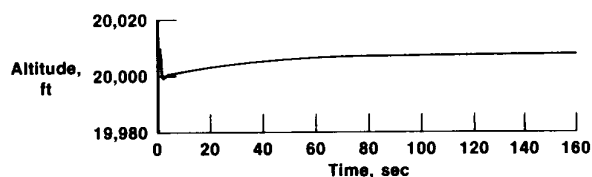
Fig. 8 Variations in altitude with time with aircraft parameters initialized to Mach 0.8 at 20,000-ft altitude

Fig. 9 Variations in velocity with time for flight condition of 20,000-ft altitude and Mach number increased from 0.5 to 0.8 at attempted state rate of 0.005 Mach per sec

Fig. 10 Variations in altitude with time for flight condition of 20,000-ft altitude and Mach number increased from 0.5 to 0.8 at attempted state rate of 0.005 Mach per sec
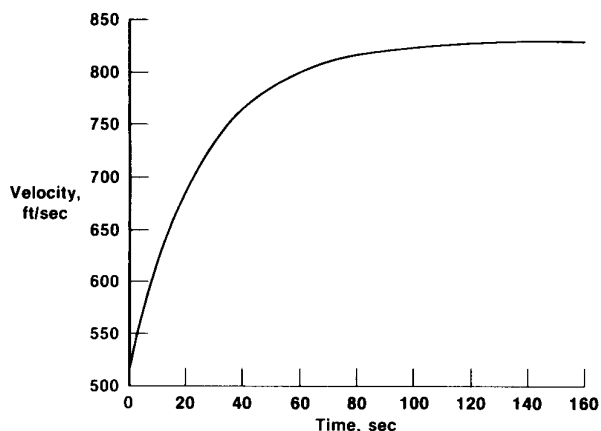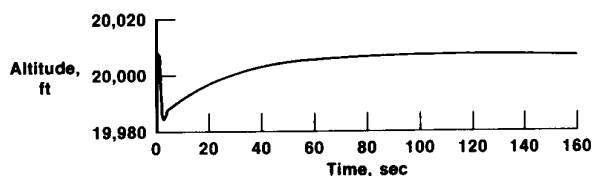
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15. Supplementary Notes

16. Abstract

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